Performance Evaluation of a Side Mounted Shuttle Derived Heavy Lift Launch Vehicle for Lunar Exploration

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Space Systems Design Lab
Georgia Tech Aerospace Eng.

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Space Systems Design Lab (SSDL)
School of Aerospace Engineering
Georgia Institute of Technology
Atlanta, GA

Author
Robert W. Thompson

Advisor
Dr. Alan Wilhite

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Robert W. Thompson*
Space Systems Design Lab
School of Aerospace Engineering
Georgia Institute of Technology, Atlanta, GA, 30332-0150

The NASA Exploration Systems Architecture Study (ESAS) produced a transportation architecture for returning humans to the moon affordably and safely. ESAS determined that the best lunar exploration strategy was to separate the launch of crew from the launch of cargo, thereby requiring two launches per lunar mission. An alternate concept for the cargo launch vehicle is a side mounted Shuttle-derived heavy lift launch. This configuration is similar to previously studied concepts, except engines and structure have been added to the External Tank (ET), making it a complete first stage. The upper stage is mounted on the side of the first stage, much like the Shuttle orbiter is mounted on the side of the ET. Like the Shuttle, solid rocket boosters (SRBs) are also used. This configuration has several performance and operational benefits over an in-line heavy lift launch vehicle.

According to the ESAS report, side mount configurations were not considered to be among the most promising configurations, and were not carried forward for further consideration within architectural options. The performance of this launch vehicle is independently analyzed, using multidisciplinary analysis techniques. Methods and tools used include launch trajectory optimization with POST, vehicle aerodynamic analysis using APAS, and weights and sizing using historically based estimating relationships. Principal trade studies performed include first and second stage propulsion (number of engines and engine type), solid rocket booster size (four versus five segment), and staging ΔV. The vehicle design that best meets the requirements for space exploration (lunar and future missions) is presented.

Nomenclature

APAS	=	Aerodynamic Preliminary Analysis System
CaLV	=	Cargo Launch Vehicle
CEV	=	Crew Exploration Vehicle
DSM	=	Design Structure Matrix
EDS	=	Earth Departure Stage
ESAS	=	Exploration Systems Architecture Study
ET	=	External Tank
LSAM	=	Lunar Surface Access Module
MLP	=	Mobile Launch Platform
NASA	=	National Aeronautics and Space Administration
POST	=	Program to Optimize Simulated Trajectories
SM	=	Service Module
SRB	=	Solid Rocket Booster
SSME	=	Space Shuttle Main Engine
STS	=	Shuttle Transportation System
TLI	=	Trans-lunar Injection
VAB	=	Vehicle Assembly Building

* Graduate Research Assistant. AIAA Student Member. Bob.Thompson@gatech.edu.

I. Introduction

The NASA Exploration Systems Architecture Study (ESAS) produced a transportation architecture for returning humans to the moon affordably and safely. ESAS recommends a “1.5 launch” architecture in which cargo and crew are launched separately in order to reduce the risk of harm to crew. The baseline launch vehicle for cargo is an in-line Shuttle-derived heavy lift launch vehicle, called the Cargo Launch Vehicle (CaLV). The in-line CaLV design, however, presents several challenges. It is more than 350 feet tall, limiting the growth of the system if required to meet increased payload requirements. This especially limits the capability to accommodate large payloads for future exploration, such as missions to Mars. The height of the launch vehicle also makes processing within the Vehicle Assembly Building (VAB) at Kennedy Space Center difficult. The in-line design also requires that the upper stage engines, which double as the propulsion elements for the trans-lunar injection (TLI) burn, be started at altitude and be restartable, modifications that could prove costly and complex in order to utilize existing engines, such as the Space Shuttle Main Engine (SSME).

A side mounted Shuttle-derived heavy lift launch vehicle would solve several of these problems. The configuration under consideration is similar to previously studied concepts like the Shuttle-C, except engines and structure have been added to the External Tank (ET), making it a complete first stage booster. This configuration is similar to the current Shuttle configuration; the upper stage is mounted on the side of the booster stage, much like the orbiter is mounted on the side of the ET. Like the Shuttle, solid rocket boosters (SRBs) are also used. Because the side-mounted configuration is more similar in configuration to the current Shuttle, processing difficulties could be lessened. Additionally, the second stage engines could be started on the launch pad, rather than being started at altitude. Finally, the system can gracefully grow to accommodate increased payload requirements for future missions.

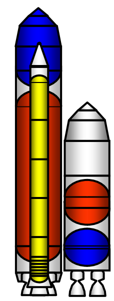


Figure 1 - Notional Launch Vehicle Configuration.

The purpose of this paper is to describe the performance of this launch vehicle, discuss the various disciplinary analyses by which it was modeled, and present recommendations for further study. Methods and tools used include launch trajectory optimization with POST, vehicle aerodynamic analysis using APAS, and weights and sizing using historical and physics based estimating relationships. Principal trade studies performed include first and second stage propulsion (number of engines and engine type), SRB size (four versus five segment), and staging ΔV. The vehicle design that best meets the requirements for space exploration (lunar and future missions) is presented.

II. Vehicle Requirements

In order to be useful for current human lunar exploration plans, the CaLV must be able to accomplish several important requirements, designed to meet performance and cost constraints on the planned lunar exploration program. The CaLV will be the largest rocket and most expensive rocket developed since the Saturn V; therefore, there is a perceived benefit to utilizing existing technologies and facilities to the maximum extent possible. For the purposes of this study, Shuttle derived technologies and facilities are used as a baseline.

As a starting point for this study, vehicle requirements were established that the launch vehicle will need to meet. These requirements were derived from the mission profile described in the ESAS Final Report[1] (Figure 2) and approximate the performance of the heavy lift launch vehicle concept advocated by that study. These requirements are listed below. Note that nominal orbital altitudes were chosen to approximate the ESAS study, and do not represent an optimal solution for this launch vehicle.

1. Use current STS facilities to the maximum extent possible
2. Deliver LSAM a 30 nmi x160 nmi x 28.5° orbit.
3. Circularize to 160 nmi x 160 nmi x 28.5° orbit.
4. Maintain orbit up to 90 days.
5. Rendezvous and dock with CEV/SM.
6. Deliver LSAM and CEV to TLI

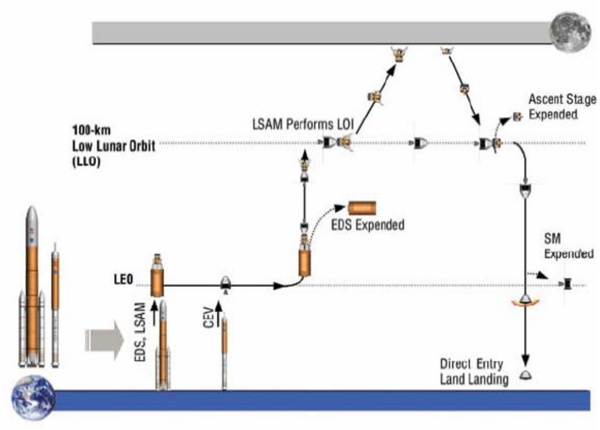


Figure 2 - ESAS Lunar Sortie[1].

It is assumed that at least one of the current Shuttle launch pads will be converted for use in the launch of the new heavy lift launch vehicles. Therefore, the use of launch infrastructure at Kennedy Space Center is also assumed, including the Vehicle Assembly Building, the three Mobile Launch Platforms, and the two Crawler-Transporters.

The CaLV must be capable of delivering the LSAM to a 30x160 nmi orbit at 28.5° inclination. This requirement becomes more difficult when coupled with the requirement of delivering the LSAM and the CEV to TLI, which means that the upper stage of the CaLV must remain partially fueled in orbit.

Because the LSAM launches unmanned in the ESAS mission profile, the upper stage of the CaLV, called the Earth Departure Stage (EDS), must wait in orbit for the CEV to rendezvous with it. The current requirement is to allow up to a 90 day wait. Thermal insulation was accounted for in the weight estimation of the EDS; however, advanced concepts for "zero-boiloff" were not considered. The performance impact of 90 days of cryogenic propellant boiloff was also not accounted for, as this issue is being explored by ongoing research programs. Rendezvous and docking is assumed to be passive on the EDS/LSAM side.

The EDS must be left with sufficient propellant to deliver the LSAM and CEV to a Trans-Lunar Injection (TLI). A nominal ΔV of 10,242 ft/s was assumed in order to maintain consistency with the ESAS launch vehicle study[1].

Table 1 - Assumed Payload Weights.

Element	Weight
LSAM	100,000 lb
CEV/SM	45,000 lb
Margin	20,000 lb

The weights assumed for each element are shown in Table 1. The margin was allocated evenly to the LSAM and CEV. The CaLV is therefore required to send 165,000 lb, or about 75 metric tons to TLI. This compares with approximately 45 metric tons (LM and CM/SM) in the Apollo program.

III. Design Methodology

A. Trade Space

The first step in the analysis of the performance of the side mount heavy lift launch vehicle was establishing the trade space that would be under consideration. This trade space includes the configuration options that will be considered. Because the performance of the launch vehicle is of interest in this study, all of the trade studies performed were propulsion related. The alternative concepts considered are shown in Table 2.

Table 2 - Alternative Concepts Considered.

Element	1	2	3	# of Alternatives
First Stage Propulsion	3 x SSME	4 x SSME	5 x SSME	3
Second Stage Propulsion	1 x J-2S	2 x J-2S	1 x SSME	3
Booster Size	4 Segment	5 Segment		2
		Total Alternative Concepts		**18**

Shuttle derived propulsion elements (the Space Shuttle Main Engine, or SSME, and Solid Rocket Boosters) were chosen because they provide good performance, and have 25 years of operational performance in the Shuttle program. The 5-segment SRB has been proposed as a performance enhancement to the shuttle, but has never been developed or put into operation. The J-2S was considered for the second stage propulsion due to its ability to restart multiple times, a requirement for the upper stage. The SSME was also considered, although it currently has no restart capability. While the J-2S has restart capability, it has not been manufactured in over thirty years. These are challenges that must be addressed in the design of any heavy lift launch vehicle that must be addressed if the SSME and J-2S are to be considered for EDS propulsion.

B. Design Structure Matrix

Once the alternative concepts have been defined, the Design Structure Matrix (DSM) was used to define the flow of information through the various disciplinary analyses of the design process. The DSM for this study is shown in Figure 3.

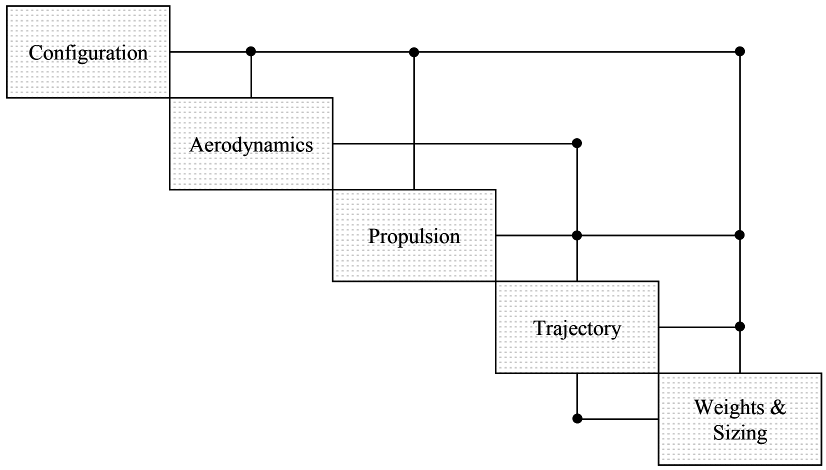


Figure 3 - Design Structure Matrix.

C. Configuration

The configuration of the vehicle has two parts: the baseline geometric configuration, and the propulsion system configuration. The first was derived from the current Shuttle configuration, and was used in calculating the aerodynamic coefficients and preliminary sizing calculations. The propulsion system configuration was derived from the 18 combinations in the alternative concepts matrix (Table 2).

D. Aerodynamics

Aerodynamic analysis was performed using APAS[2] for vehicle modeling, and HABP for hypersonic aerodynamic analysis. The launch vehicle was modeled as four parts: The first stage, second stage (EDS), and the two SRBs. The aerodynamics of three configurations were modeled: launch, after SRB separation, and after first stage separation. These configurations are shown in Figure 4. The cross sections and lengths of the final launch vehicle elements varied considerably from the initial APAS model (as will be shown in later sections); however, these changes do not have a significant impact on the aerodynamic coefficients as outputted by APAS into POST, the trajectory code. Instead, the reference areas and stage lengths are updated within the trajectory code, and the coefficients are kept constant.

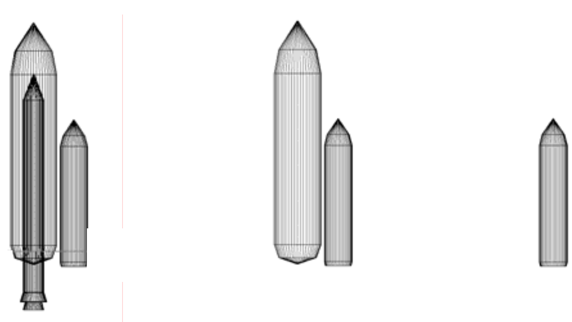


Figure 4 - APAS model of the CaLV in launch, post SRB separation, and post staging configurations[2].

E. Propulsion

The propulsion subsystems were modeled using both historical data and a conceptual rocket engine sizing tool, REDTOP-2[3]. The four propulsion systems modeled were the Space Shuttle Main Engine (SSME), the J-2S, the four-segment SRB, and the five-segment SRB.

The SSME (also known as the RS-25) was modeled using historical information. The SSME was considered to be unchanged from its current Block II configuration. The parameters used for the SSME are shown in Table 3.

Table 3 - Space Shuttle Main Engine (SSME) Parameters[3]

Parameter	Value	Units
Vacuum Thrust	500,000	lb
Vacuum Specific Impulse	452	s
Exit Area	45	ft^2
Weight	7,000	lb

The J-2S is a derivative of the J-2 engine produced by Rocketdyne in the 1960s. The J-2S (the S is for "simplified") is a tap-off cycle variant of the J-2, which was a gas generator cycle[4]. The nozzle area ratio has been increased for better specific impulse. This engine was modeled using the REDTOP-2 code with an increased expansion ratio of 101; this improves engine performance, but lengthens the engine to about 16 feet. The parameters used for the J-2S engine are shown in Table 4.

Table 4 - J-2S Derived Engine Parameters[3].

Parameter	Value	Units
Vacuum Thrust	275,000	lb
Vacuum Specific Impulse	451	s
Exit Area	80	ft^2
Weight	4800	lb

Both 4-segment and 5-segment SRBs were modeled for use in this study. The thrust profile of the 4-segment SRB is the same as that of the current Shuttle SRB. The 5-segment SRB thrust profile is derived from a design intended to increase Shuttle performance. The relevant parameters for each are shown in Table 5 and Figure 5.

Table 5 - Four and Five Segment SRB Parameters.

Parameter	4-Segment		5-Segment	
Vacuum Thrust	Variable		Variable	
Vacuum Specific Impulse	265	s	265	s
Exit Area	130	ft^2	130	ft^2
Burnout Weight	185,000	lb	220,000	lb
Gross Weight	1,293,000	lb	1,630,000	lb

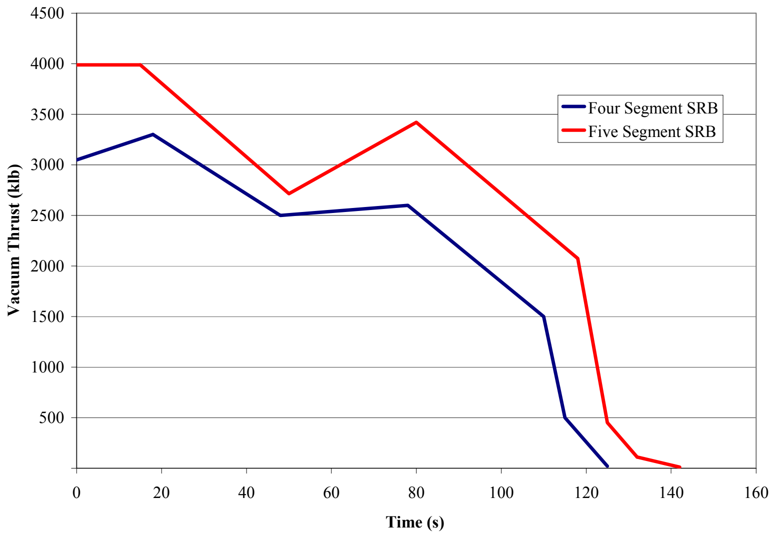
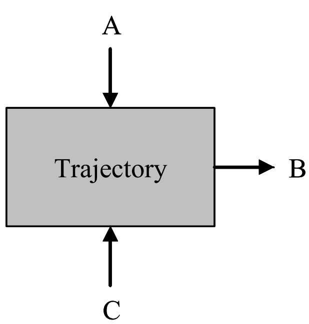


Figure 5 - Four and Five Segment SRB Thrust Profile.

F. Trajectory

The Program to Optimize Simulated Trajectories (POST)[5] was used to perform the trajectory simulation. Because of the complexity of the flow of information in and out of POST during the sizing process, a simple schematic of this is shown in Figure 6. Information from the aerodynamic and propulsion analyses was brought forward and used as input for the trajectory analysis. An initial guess was made of the stage dry weights; the trajectory was then run using a projected gradient optimization. The outputs to the weights and sizing analysis are the propellant weights of each stage, which returns the stage dry weights to the trajectory analysis. Because of this feedback loop, iteration was required to "close" the performance of the vehicle. This was done manually, and typically required 5 to 6 iterations to converge to the final solution.



A – Feed Forward Inputs:	Aerodynamic Coefficients
	Propulsion Thrust, Isp
B – Feed Forward Outputs:	1st and 2nd Stage Propellant Weight
C – Feed Back Input:	1st and 2nd Stage Dry Weights

Figure 6 - Trajectory Analysis Data Flow.

The trajectory optimization was constrained to meet the final conditions specified by the desired orbit at engine cut-off, and the desired mass at the end of the launch trajectory. These constraints are shown in Table 6.

Table 6 - Trajectory Optimization Constraints.

Constrained Parameter	Value
Altitude	78 nmi
Flight Path Angle	1.0168°
Inertial Velocity	25,708 ft/s
Final Weight	Feedback from Weights

The constraints on altitude, flight path angle, and inertial velocity place the EDS and launch payload on a 30 x 160 nmi orbit, inserting at 78 nmi. This initial orbit is the same as that used by the ESAS in-line CaLV; the same orbit was used in order to be able to compare performance numbers between the vehicles.

G. Weights and Sizing

An important part of the performance sizing loop is the Weights and Sizing analysis. The inputs for this analysis come from the vehicle configuration, propulsion options, and trajectory. As shown in Figure 6, the propellant weights required for launch vehicle are fed into the weights and sizing spreadsheet, which then calculates the dry weight of the vehicle parametrically. This sizing is based on a database of mass estimating relationships from several sources, which have been compared against the Shuttle for verification of their accuracy[6]. These mass estimating relationships are typically parametric regressions of historical data.

Included in the weight estimation is the estimation of dry weight margin. Each subsystem was assigned a percentage of dry weight margin based on the heritage of the system. Systems that are derived from already designed an operational systems are given less margin than systems that will have to be developed from scratch. The dry weight margin percentages are shown in Table 7.

Table 7 - Dry Weight Margin based on Level of Heritage.

Level of Heritage	Margin Applied
New	15%
Moderate Modifications	10%
Minor Modifications	5%
Existing	0%

IV. Vehicle Closure Results

As a result of the analysis of the propulsion configurations considered (shown in Table 2), certain options were eliminated due to the inability to close a vehicle with those options that met the lunar exploration requirements. The eliminated propulsion options are:

1. 3 x SSME on the first stage.
2. 1 x J-2S on the EDS.
3. 4 segment SRBs.

These options were found to provide insufficient thrust when combined with each other and with other options. They are therefore considered to be infeasible for a heavy lift launch vehicle for lunar exploration within the ESAS 1.5 launch architecture. A single J-2S on the upper stage was eliminated because it provides an insufficient T/W at staging to accomplish the mission. Due to the lower thrust level, the burn time must increase, which increases the amount of propellant required. The four segment SRBs and three SSME options are eliminated because they provide insufficient performance for lunar exploration. To verify this, the vehicle in this configuration was compared to the Shuttle, which utilizes these propulsion elements. The Shuttle system places approximately 300,000 lb in LEO (including the ET). The comparable side mounted CaLV has a similar capability to LEO. However, based on the requirements of the TLI burn and the payload requirements, around 380,000 lb are needed in LEO. Therefore, it can be concluded that a vehicle with a Shuttle-like propulsion configuration does not have the performance necessary to act as the heavy lift launch vehicle in the ESAS architecture.

The closure results indicate that a four or five SSME first stage propulsion configuration is possible. With a five SSME configuration, first stage engine out capability could be possible. The other configuration options considered for the EDS propulsion (2 x J-2S versus 1 x SSME) were very close in performance.

Based on these results, a baseline side mount heavy lift configuration was chosen. A rendition of the closed vehicle is shown in Figure 7 .

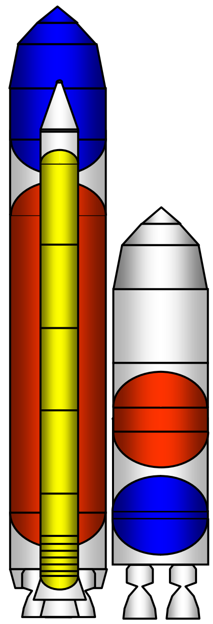


Figure 7 - Closed Launch Vehicle Configuration.

The closed configuration utilizes five SSMEs on the first stage, two five segment SRBs, and two J-2S engines on the EDS. The five SSMEs on the first stage were chosen due to the option for engine-out capability, and due to the potential for future growth of the launch vehicle. Because the four segment SRBs did not have the thrust necessary for this vehicle, the five segment SRBs were used in the baseline. Finally, the two J-2S engines on the EDS were picked rather than the SSME because the J-2S has the capability to be restarted multiple times. The weights of the two first stage and EDS are shown in Table 8 and Table 9. The estimated dimensions of the baseline concept are shown in Figure 8 and Figure 9.

Table 8- EDS Weights.

Element	Weight
Dry Weight w/ Margin	45,000 lb
Residuals	2,000 lb
Launch Payload (LSAM)	110,000 lb
Rendezvous Payload (CEV)	*55,000 lb*
TLI Propellants	217,000 lb
Circularization Propellant	6,500 lb
Ascent Propellants	170,000 lb
Initial Weight in LEO	374,000 lb
Total Weight to TLI	212,000 lb
Gross Weight at Liftoff	**551,000 lb**

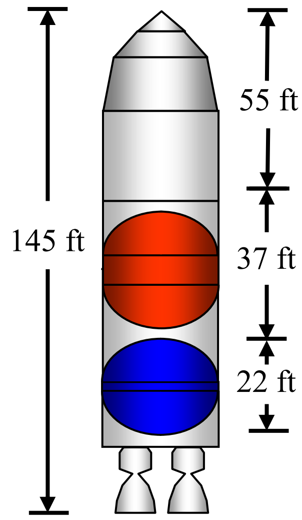


Figure 8 - EDS Dimensions.

Table 9 - First Stage Weights.

Element	Weight
Dry Weight w/ Margin	184,000 lb
Residuals	19,000 lb
Second Stage (EDS)	551,000 lb
Ascent Propellants	**2,120,000 lb**
Fuel	305,000 lb
Oxidizer	1,815,000 lb
Max Liquid Stage Weight	2,875,000 lb
Five Segment SRBs	3,300,000 lb
Gross Liftoff Weight	**6,175,000 lb**

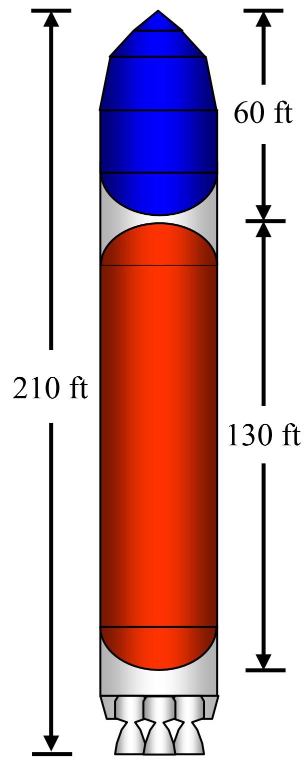


Figure 9 - First Stage Dimensions.

The core stage is approximately sixty feet longer than the Shuttle ET, from which it is derived. A trade study of staging ΔVs was performed, and it was found that the best staging point is at an ideal ΔV of 23,500 ft/s, after about 311 seconds for this trajectory. Note that this ideal ΔV does not include drag and gravity losses. An altitude versus time plot of the trajectory of the baseline launch vehicle is shown in Figure 10. The axisymmetric acceleration is shown in Figure 11. Plots of vehicle weight versus time and inertial velocity versus time are shown in Figures 12 and 13 respectively.

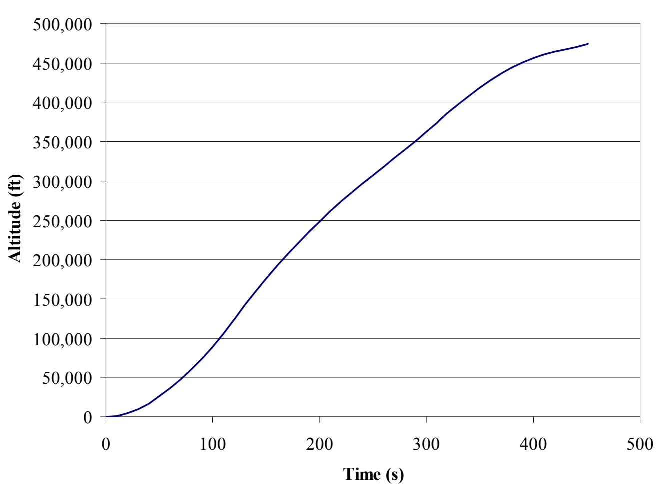


Figure 10 - Altitude versus Time.

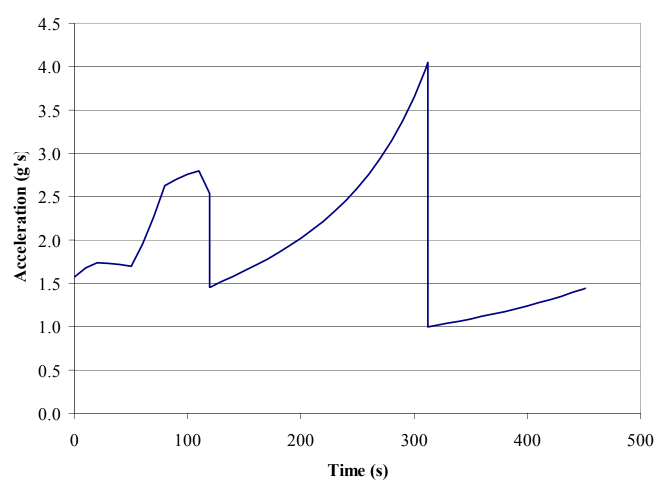


Figure 11 - Acceleration versus Time.

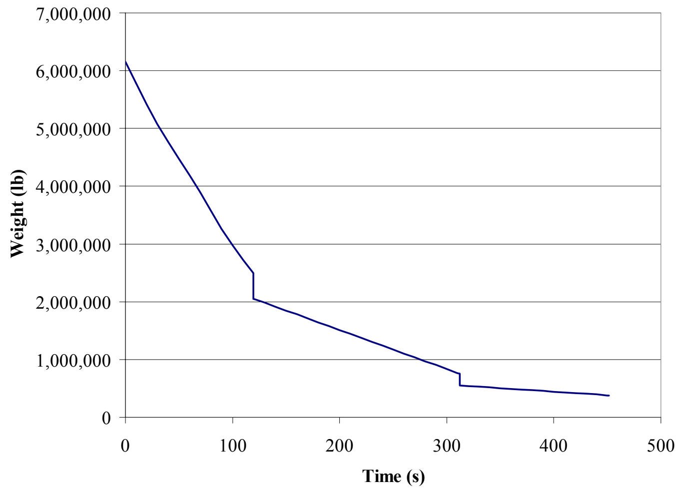


Figure 12 - Gross Weight versus Time.

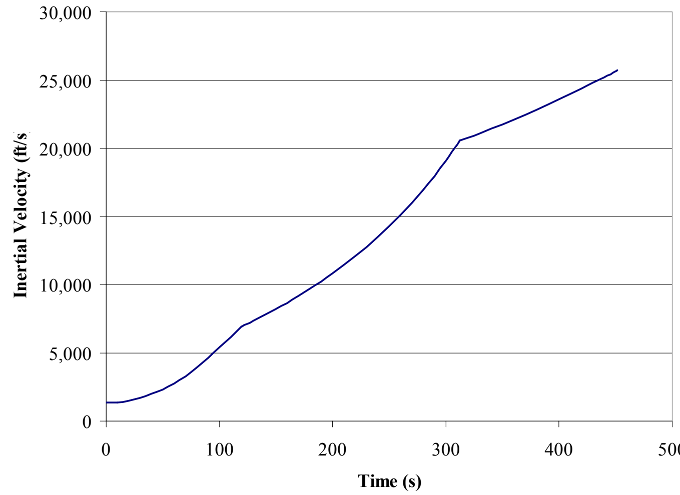


Figure 13 – Inertial Velocity versus Time.

The altitude at orbital insertion is approximately 78 nmi. The time to orbit is 451 seconds, and the maximum g's are 4.0 g's at the end of the burn of the first stage. A max-g limit anywhere between 3.0 and 6.0 g's can be accommodated with minimal change in performance; this is accomplished through throttling the main engines, and impacts the total burn time.

The stage dry weight and gross weight comparisons to the ESAS in-line CaLV are shown in Table 10. The stage fueled weights do not include payload. The dry weights are similar because, although less propellant is required, there are additional hardware systems (crossover feed lines and side-mounting hardware) that are used on the side mounted configuration.

Table 10 - Comparison of stage dry and fueled weights to the ESAS in-line CaLV.

Element	ESAS In-line[1]	Side Mount
First Stage Dry Weight	194,997 lb	184,000 lb
EDS Dry Weight	44,118 lb	45,000 lb
First Stage Fueled Weight	2,409,997 lb	2,320,000 lb
EDS Fueled Weight	541,238 lb	440,500 lb
Gross Liftoff Weight	**6,393,975 lb**	**6,175,000 lb**

V. Future Work

In order to compare the results of this vehicle study directly to the results of the launch vehicle analysis done by ESAS, many of the same assumptions were used, such as propulsion choices, trajectory parameters, orbits, as well as dry weight margins and tank assumptions (reserves, residuals, ullage, etc.). The validity of these assumptions should be investigated, documented as to their basis in previous vehicles and missions, and the sensitivities to changes in these assumptions on vehicle performance should be noted. There are also other propulsion options that could be investigated. The performance of other LOX/Hydrogen engines, such as the RS-68 and RL-10 should be investigated. While only liquid oxygen and hydrogen propulsion systems were considered in this study because they have the greatest heritage from the Space Shuttle program, there is Apollo heritage for using Kerosene/RP-1 as a fuel.

In order to confirm the feasibility of this launch vehicle configuration concept, a detailed structural analysis of this launch vehicle should be performed. It is possible that the dry mass weights of the stages (especially the first stage) will increase when the structural analysis is incorporated into the performance DSM. A sensitivity study to stage dry weight margin will be performed to quantify the degree of dry weight growth that can be accepted by this configuration.

In addition to researching the propulsion options, some attention should be paid to engine-out options. The baseline configuration presented in this study can deliver its payload into orbit after losing one main engine, but the upper stage requires a longer circularization burn. Whether or not this is possible depends on how margin is allocated in the vehicle, which should be a topic of further study. In examining engine-out, the reliability of the launch vehicle should be estimated, in order to assess the potential pay-off in developing that capability. Finally, cost estimation and a more detailed analysis of the operations of this vehicle should be completed.

VI. Conclusions

The side mount CaLV concept is one that shows promise for lunar exploration. It is capable of accomplishing all of the goals for lunar exploration required by the currently baselined 1.5 launch architecture for returning humans to the moon. This launch vehicle concept shows advantages over the in-line concept. The side mount concept shows a small performance improvement due to the additional engines that are burning through the whole trajectory. It shows potential for growth in future missions and, like the in-line CaLV, has a payload capacity to LEO of over 150 metric tons. Its shorter length and Shuttle-like configuration will ease the transition from current Shuttle operations. In the case of a failure of an engine on the EDS, the launch can be aborted. If such an EDS engine failure occurred on the in-line launch vehicle, the mission would fail.

There are, however, some disadvantages to the side mount configuration. The cross-feeding of propellants adds complexity to the design. The upper stage engines also have a significantly longer burn time, increasing their probability of failure. Those same engines have nozzles that are optimized for in-space operation, so there are thrust losses until atmospheric pressure lessens, approximately 90 seconds into the launch trajectory. The impacts of these disadvantages on life cycle cost and vehicle reliability should be studied as future work progresses on this concept. As exploration systems studies continue, this concept should be considered as an alternative to the in-line launch vehicle concept.

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